THE PIONEER SPACECRAFT

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On December 16, 1965, with the successful launch of Pioneer 6, National Aeronautics and Space Administration* began a systematic exploration of interplanetary space between 0.8 and 1.2 AU. This new series of four Pioneer Spacecraft flights at regular intervals is a continuation of the scientific spacecraft series interrupted since Pioneer 5 reached its maximum communication range on June 26, 1960. The basic purpose of this series is long-term monitoring of interplanetary phenomena while well outside Earth's influence and at varying distances between 0.7 and 1.2 AU from the Sun.

This paper describes the design of the Pioneer Spacecraft as carried out at TRW Systems under the management of Ames Research Center, NASA, and the scientific objectives that are being achieved with the first flight. It is hoped that this paper will simultaneously demonstrate the wider applicability of this generic class of which Pioneer is but one example. The paper will present the original design constraints, objectives, and major problems, followed by a description of how they were satisfied in terms of the present design. Then a description will be given of the performance of Pioneer 6, the first flight in the series, and of its payload of scientific instruments. Unfortunately, at the time of writing this paper there has not been sufficient time for the various scientific investigators to analyze and publish their results (Ref. Dr. F. Scarf "Wave Particle Interactions in the Solar Wind", to be presented at the Symposium on Exploration of the Moon and Interplanetary Space, Deutsche Gessellschaft für Raketentechnik und Raumfahrt e.V. München, April 22, 1966). Consequently, only a qualititative discussion of scientific observations will be given here. The paper will conclude with a mention of possible future applications for Pioneer that could be achieved with minor design changes to the present spacecraft.

Design History

Objectives established in 1963 guided major design decisions on Pioneer. They can be categorized as both scientific and programmatic.

*Work described in this publication was performed under NASA contract NAS2-1700.

The principal scientific objectives follow.

a) Omnidirectional scanning capability, particularly in the plane of the ecliptic.

b) Continuous experiment sampling.

c) Maximum quantity of data consistent with weight and power limitations.

d) A stable, predictable scientific instrument environment, including slowly varying temperatures, low electromagnetic interference, and low residual magnetic fields.

e) Maximum heleocentric radius variations consistent with booster and communication range capabilities.

Programmatic objectives primarily stemmed from a basic desire to maximize the cost effectiveness of each flight. They are summarized briefly, following.

a) Utilization of low cost launch vehicles.

b) A high probability of long life with no intrinsic end of life.

c) All spacecraft to be capable of operation at varying radial distances from the Sun without modification.

d) Maximization of the scientific yield per flight, in terms of quality and quantity of data.

These objectives led directly to the design of Pioneer as a light-weight, spinning spacecraft. (See Figure 1). It was felt the use of spin for attitude stabilization would decrease the total spacecraft weight for a given scientific payload and at the same time increase lifetime. The maintenance of spin over a long period of time seemed feasible, since the perturbing torques in interplanetary space are small compared to the near-Earth environment. Perhaps equally important, however, spin allows scientific instruments to have spatial resolution without special scanning platforms.

In order to select the optimum direction of scan for the experiments and to increase the communication range, it was decided to orient the spacecraft spin axis perpendicular to the ecliptic and to utilize a communication antenna with a flat, pancake, symmetric pattern that would continuously illuminate the Earth throughout mission life. With this orientation, no active antenna pointing capability is necessary as long as the spacecraft orbit remains sensibly close to the ecliptic plane.

Having the spin axis perpendicular to the ecliptic yields additional advantages. The traditional heavy solar paddles can be eliminated and replaced with a cylindrical array wrapped around the spacecraft equipment. In addition, thermal control is more easily achieved when the spacecraft orientation is known. The excess heat can be radiated to free space from either end of the cylinder, and thermal control design is thereby to a large extent independent of radial distance from the Sun.

It is well to say at this point that some of the original objectives were mutually contradictory and, therefore, a balance among them was necessary. The desire to have high data rates implied a large solar array for the RF power required and hence more total weight or less scientific instrumentation. However, the decision by NASA to use the thrust-augmented Thor Delta as the launch vehicle determined the maximum solar array diameter and length, and total spacecraft weight. Having determined these two factors, the maximum power available for communication at 1.2 AU from the Sun was specified within fairly narrow limits.

Another major design consideration, worthy of note here, was the decision to eliminate the necessity for a duty cycle or battery assist operation. Experience indicates that batteries have often been the first item to fail on long missions. Therefore, Pioneer has been designed to operate continuously from solar array power only after fairing removal during launch and proper orientation over the region of 0.8 to 1.2 AU from the Sun. The design life of six months, the communication range from Earth at this time, the limited solar array area, and the antenna design must be compatible. Satisfying these factors determined the major characteristics of the communication system.

During the actual development phase, a major problem turned out to be holding the spacecraft weight close to the original predictions, since the booster capabilities were almost completely utilized by the original design weights. Table 1 shows the weight breakdown finally achieved, with an historical comparison to Pioneer 5. The total weight growth between conceptual design and final launch weight was held to less than five percent. One general design approach used to maintain a low spacecraft weight was the conscious elimination of as much onboard decision making as possible. This design approach, of course, magnifies the ground control function and requires careful attention to the design of a safe, reliable command system (both on the ground and in the spacecraft) to handle the relatively high volume of command activity required.

Table 1. Pioneer Weight Allocation

	Pioneer 5		Pioneer 6	
	Weight (pounds)	Percentage of Total	Weight (pounds)	Percentage of Total
Total Spacecraft Weight	95.0		140.0	
Scientific Experiments	9.5	10	35.0	25
Communications, Timing and Data Handling	28.5	30	35.0	25
Electrical Power	33.0	35	28.0	20
Structure, Pyro, Cabling and Thermal Control	23.8	25	33.6	24
Attitude Control & Propulsion	0	0	8.4	6

A second major problem area involved magnetic cleanliness. In order to achieve the design goal of a residual magnetic field of less than one gamma (1 gamma = 10^{-5} gauss) at 80 inches from the spin axis where the magnetometer sensor is located, intensive effort was required early in the design phase. Wherever possible, all ferro-magnetic materials that might be present in electronic parts or welding and structural materials had to be eliminated. This meant that parts normally used in spacecraft design such as tuning forks, latching relays, and circulators for RF switching had to be replaced by alternate devices. Then, at the next level of design, current loops had to be eliminated in individual boxes and in the overall spacecraft power circuitry by careful consideration of grounding methods. The solar array, of course, presented a special problem. The magnetic fields that would be normally present were eliminated by arrangement of internal wiring to achieve self-cancellation of magnetic fields at the surface of the array. In addition to the magnetic design activity, an intensive and carefully regulated test program was necessary to be certain that no residual contamination from ferro-magnetic tools used in forming plastics or inadvertent material substitutions were present. As the final test, the total spacecraft was put into an essentially field-free region of Helmholtz coils designed to cancel the Earth's field.

While inside these Helmholtz coils the residual spacecraft field was mapped under all operating modes. These test results showed that the maximum background field from the spacecraft that would be measured by the magnetometer on the boom in flight was less than 0.25 gamma.

Spacecraft Configuration

The spacecraft is a drum-shaped container, 35 inches high and 37 inches in diameter, enclosing the majority of experiments and spacecraft equipment (see Figure 2). The experiment sensors are located in the circular band dividing the solar array. The antenna projecting from the bottom of the spacecraft consists of two antennas at 49.8 and 423.3 mc for receiving signals from Earth for the Radio Propagation Detector, one of the experiments.

At 120-degree intervals around the cylinder are three deployable booms which carry respectively, a magnetometer sensor, the gas jet for orientation maneuvers, and a wobble damper to eliminate nutation after orientation maneuvers have been completed.

The circular equipment platform housing most of the electronics equipment is aluminum honeycomb, below which are mounted thermally actuated louvers to dissipate any excess heat from the platform.

Extending from the top of the cylinder is a four-foot antenna mast containing the high-gain (11db) Franklin array, as well as two omnidirectional antennas for near-Earth down-link communications and up-link out to maximum range. This antenna mast also serves as a support for the three booms when they are folded in launch configuration. Also attached to the antenna mast is a small flat plate, or solar sail, which places the center of solar pressure close to the center of gravity of the spacecraft. This eliminates any torque by solar illumination which would cause nutation of the spin axis and tilt the spacecraft spin axis away from perpendicularity to the ecliptic plane.

Experiments

Pioneer 6 carries six experiments for measurements of electrons, protons, alpha particles at various energies in the vicinity of the spacecraft, the total electron density over long path lengths, and interplanetary magnetic fields.

The principal investigator for magnetic field measurement is Dr. N. F. Ness, Goddard Space Flight Center, National Aeronautics and Space Administration. The magnetometer, whose sensor is located at the end of one boom, is sensitive to ±0.25 gamma and has a dynamic range of ±64 gamma. Since the quiescent interplanetary field is about 5 gamma, this instrument is sufficiently sensitive to measure the steady-state as well as disturbed fields during intense solar activity. The active element of the magnetometer is a single-axis sensor oriented at 54 degrees, 45 minutes to the spin axis. As a result of this orientation, during one revolution of the spacecraft all three components of the interplanetary field are measured. Upon ground command, the sensor can be rotated 180 degrees to cancel draft effects associated with the sensor or electronics. The sensor can also be commanded into a calibrate mode corresponding to a standard 10 gamma field.

Two cosmic ray experiments are carried in Pioneer 6. The first, whose principal investigator is Dr. J. A. Simpson, Fermi Institute, University of Chicago, measures proton and alpha flux in the energy range of 1 to 200 mev with a 60 degree viewing angle. The second, whose principal investigator is Dr. K. G. McCracken, Graduate Research Center of the Southwest, measures protons between 5 and 90 mev, and alpha particles between 130 and 360 mev. This instrument is designed to detect anisotropy in the primary cosmic radiation. Both of these experiments should provide information on the interaction of the solar and interstellar magnetic fields.

The radio propagation experiment, whose principal investigator is Dr. V. R. Eshleman, Stanford University, consists of two phase-lock receivers operating at 49.8 and 423.3 mc. Signals at these frequencies with known phase relationships are transmitted from the Earth from a 150-foot antenna located at Stanford University and the phase and group delay of the two frequencies is compared on the spacecraft and transmitted back to Earth via the spacecraft telecommunication system. Measurements made by this method allow calculation of the total number of electrons present and their time variation along the line-of-sight between the spacecraft and Earth.

Two plasma probes measure the flux, incident angle, and energy spectrum of the solar plasma. The first of these, whose principal investigator is Dr. H.S. Bridge, Massachusetts Institute of Technology, consists of a Faraday cup and grid collector which measures fluxes of electrons in the energy range of 500 to 2500 ev and protons in the range 40 ev to 10 kev in ranges of 16 steps. It can measure flux densities of between $4 \cdot 10^5$ to $4 \cdot 10^9$ particles/cm^2/sec with a resolution of 10 degrees in the ecliptic. The second plasma probe, whose principal investigator is Dr. J.H. Wolfe, Ames Research Center, National Aeronautics and Space Administration, is an electrostatic particle detector measuring fluxes of electrons in the range of 2 to 500 kev and positive ions in the range of 0.20 to 10 kev with a dynamic range between $0.5 \cdot 10^5$ and 10^8 particles/cm^2/sec. This experiment has an angular resolution of 3 degrees in the ecliptic and divides the energy spectrum into 7 electron steps and 15 proton steps. It can sample in all directions except for a 20 degree cone centered around each spacecraft pole pointing toward the ecliptic poles.

Spacecraft Description

Table 2 summarizes important characteristics of the spacecraft in addition to the experimental payload already described. Of particular interest, as indicated by the table, is the attempt to utilize redundancy wherever reliability calculations

Table 2. Spacecraft Characteristics

Communications - 2200 mc

Receivers (2): Phase lock with ground station for 2-way doppler, or non-coherent for standby operation, 12 cps bandwidth

Command Decoders (2): 57 different commands available

Solid State Driver (1): 0.05 watts to antenna at 2200 mc

Traveling Wave Tube Amplifiers (2): 8 watts to antenna at 2200 mc

Antennas (3): 11 db gain, 2 omnidirectional maximum 5.25 watts radiated

Coaxial Switches (5)

Data Handling

Memory Capacity: 15.10^3 bits

Different Message Formats: 3 scientific, 1 engineering

Different Modes of Operation: 4 (2 storage, 2 real time)

Different Bit Rates: 215, 256, 64, 16, 8 bits per second

Power

Solar Cells: 10,300 N-on-P silicon cells

Launch Power: 2 amp hour rechargeable battery

Total Solar Power Available: 60 watts at 1.2 AU, 83 watts at 1.0 AU, 100 watts at 1 AU

Experiment Power Available: 15 watts at 1.2 AU, 28 watts at 1 AU, 55 watts at 0.8 AU

Equipment Converters: (2)

Traveling Wave Tube Converters: (2)

Thermal

Equipment Platform: 40-80°F between 1.2 to 0.8 AU from the Sun

Orientation

N_2 cold gas, 0.5 degree pointing accuracy

240 degrees of total rotation available

indicated that this would be desirable to insure the six-month lifetime goal. For example, as noted in the functional block diagram (Figure 3), both receivers and command decoders are redundant as well as the traveling wave tubes (TWT) and their power converters. In addition, by choosing appropriate switch positions from ground command, either the driver alone or either one of the TWT's can be switched to high-gain or omnidirectional antennas. As another example of redundancy, the electronics for the orientation logic and sun sensors have been made quad-redundant at the part level to ensure success of this critical maneuver. As noted, the power distribution system uses redundant voltage converters for these critical functions. It should be pointed out in addition that no use is made of batteries after launch; the battery is simply left floating on the main bus to be slowly recharged throughout the mission for possible use in emergencies. The power distribution system also contains an undervoltage control that in an emergency shuts off nonessential power loads such as the experiments and treveling wave tubes. This occurs automatically whenever a short or other fault lowers the main bus voltage below 23 volts. After the fault has been eliminated, the units are turned on by ground command individually. Other portions of the spacecraft power circuitry are protected by fuses where redundant units are available in case of failures.

As can be seen in Table 2, the data handling system has been designed, as far as possible within weight limitations, to maximize the efficiency of experiment data collection. There is sufficient flexibility in data rates to optimize the data collection rate to obtain the maximum allowable at each communication range. In addition, format of the scientific message can be changed to allow for different modes of experiment operation as the bit rate (and therefore sampling rate) is decreased. Also, data sampling and storage at regular, predetermined intervals is provided for during periods when ground stations are not monitoring the spacecraft, so that no large gaps in experiment data will exist in the event of lack of coverage from the ground.

The orientation system consists of five Sun sensors, control electronics, a pressure vessel with N_2 as the working gas, associated regulation valves, and a nozzle at the end of one boom. The Sun sensors (see Figure 4) located around the mid-section of the spacecraft have differing Sun shades depending on their function. Sensors A and C, with viewing angles as shown in Figure 5, control the automatic Step I maneuver that tilts the spin axis normal to the spacecraft Sun line. As either sensor A or C sees the Sun, the gas jet is fired for a portion of a revolution until both sensors view the Sun and the maneuver automatically terminates. By ground command, the orientation logic selects either sensor B or D to control the Step II maneuver. Since sensors B and D are placed 90 degrees away from A and C, they will rotate the spacecraft around an axis normal to the initial rotation axis and therefore not disturb the solar illumination incidence angle. The sense of rotation is pre-selected by allowing either B or D to control the gas firing. The Step II maneuver is terminated when ground observations indicate the spacecraft antenna pattern is properly illuminating the Earth. By knowing the actual trajectory flown, the spacecraft attitude can thereby be inferred and biased in any direction by rotating around the spacecraft Sun line as an axis. In this manner, if the spacecraft orbit is out of the ecliptic, spacecraft attitude can be pre-positioned to yield an optimum antenna pattern at more distant communication ranges. The fifth sensor, E, provides a pulse during each spacecraft revolution, used by the scientific equipment for determining the direction of observation at any instant of time.

Pioneer 6 Flight Summary

Pioneer 6 was launched on December 16, 1965, by a thrust-augmented Thor with an improved Delta second stage and injected into an escape trajectory after a 16-minute coast over the South Atlantic. The heleocentric orbital elements, which turned out to be extremely close to the planned elements, are shown in Table 3.

Table 3. Heliocentric Trajectory Elements

Semi-major axis	=	134,485,990 km
Eccentricity	=	0.094190951
Inclination	=	0.16953981 deg
Perihelion Distance	=	121,818,630 km (0.815 AU)
Orbital Period	=	311.33001 days

As can be seen from the orbital elements, and the plot in Figure 6, the spacecraft is slowly advancing with respect to the Earth at a rate which will return it into the vicinity of Earth in about six to seven years. The orbit inclination is sufficiently small (<0.2 degrees with respect to the ecliptic) so that the spacecraft antenna pattern will continue to intersect the Earth throughout mission life. Perihelion passage at 0.814A occurs on May 18, 1966, at which time the spacecraft will be about 40 million miles from Earth.

Immediately after injection and separation from the third stage, the booms and antenna deployed and the traveling wave tube was turned and switched on to the omnidirectional antenna, all automatically. This provided adequate radiated power so that the DSN station at Johannesburg acquired the spacecraft as soon as it appeared over the horizon. Also, as the spacecraft separated from the third stage, the orientation system automatically oriented the solar array perpendicular to the spacecraft Sun line to provide maximum solar power. The NASA ground controllers at Johannesburg and Pasadena quickly determined that the spacecraft was operating properly and began turning on each experiment. By four hours, all six experiments were operating and collecting data as the spacecraft traversed the transition region between the interplanetary and Earth's magnetic field at about 10 Earth radii. As a consequence of the 512 bits/sec data rate, the magnetometer and plasma experiments obtained good data through this interesting region. Within three hours, the DSN stations and orbit computation facility, both under the direction of the Jet Propulsion Laboratory at Pasadena, California, had determined the trajectory accurately enough so that, combined with ground station signal strength measurements, the actual spacecraft attitude achieved could be ascertained within several degrees.

On the second day of flight, from the DSN station at Goldstone, the spacecraft was rotated by ground command through one-third of a degree per step around the spacecraft Sun line axis. As the spin axis approached perpendicularity to the spacecraft Earth line, the TWT was switched to the high-gain, directional antenna. By observing the received signal strength at the ground station, together with previous knowledge of the antenna pattern, the spacecraft was brought within one degree of perpendicularity. The Step II maneuver required a rotation of about 75 degrees by ground commands. Since the spacecraft orbit achieved is sufficiently close to the ecliptic, no further orientation maneuvers will be necessary to maintain the proper antenna pointing throughout the six-month flight.

During this early mission phase, as may be expected, command activity was quite high. Over 500 commands were sent and executed prior to completion of orientation on the second day. Since that time, over a period of four months, approximately 2000 commands have been sent and executed without a failure.

From the engineering standpoint, the flight has been routine, with no failures detected or redundant units required during the first four months. As can be seen from Figure 7, the maximum bit rate of 512 bits per second was maintained from launch to early in March, allowing the experimenters to obtain maximum temporal resolution during this period. Figure 7 shows the actual signal strength measured at the various ground stations, compared to the curve of signal strength predicted for nominal performance before launch. As can be seen, both the spacecraft and ground stations appear to be exceeding the performance expected. The points indicated as transitions from 512 to 256, etc., bits per second are based on a bit error rate, P(E), of 1 in 10^{-3}. When the error rate grows significantly larger than this, danger exists of introducing spurious experimental data points. The slight bulge in the signal strength curve noticeable around 70 days is a result of the small heliocentric orbital inclination, which causes the spacecraft antenna pattern to drift about one degree, with respect to Earth, throughout the flight.

Starting about the 10th of January, solar activity increased. During subsequent weeks, observations were made during numerous Class 1 and 2 and several Class 3 solar flares. Unfortunately, at the time of writing this paper, there has not been time for scientific results to be analyzed sufficiently to state quantitative results. In general, however, early indications show that the interplanetary plasma has a more complex structure than observed heretofore. Particles in different energy bands and of different species arrive at the spacecraft from different directions. In other words, the fine structure of the interplanetary plasma is now being observed with definite indications of turbulance and other complex structure not previously noted on interplanetary flights. It has also been observed that the spacecraft has collected a cloud of low energy electrons, probably photoelectrons from the solar array, which are easily detectable as the instrument sensors rotate within this cloud.

It is expected that future Pioneers will carry experiments of similar types, but with different dynamic ranges and sensitivities, matching increasing flare activity as the years of maximum solar activity are approached.

Applications of Pioneer

The basic limitations of the present Pioneer are insufficient electric power beyond 1.3 AU from the Sun, excessive heating of the solar array at 0.6 AU from the Sun, and a maximum communication range (at 8 bits/sec) of about 50 million miles. Within these limitations, however, there are several interesting additional missions that could be performed.

Interest in the nature of circadian rhythm of biological organisms (Ref. 1 and 2) has led to the desire to place some well calibrated species such as pocket mice, cockroaches, fruit flies, red algae, and potatoes into an environment well away from any Earth influence. It is also desirable that they be maintained in a 1-g field during these observations. Both of these requirements can be satisfied on Pioneer by proper positioning relative to the spin axis. If such an experiment were conducted, biologists feel they could distinguish between a built-in clock mechanism, perhaps at the cellular level, or organic sensors as the determinants for establishing these circadian rhythms. In addition, if the rhythms are controlled by sensors, something may be learned of how the rhythms are disturbed and how they are reestablished, if at all, when the external environment is altered.

The communication range for interplanetary missions will be increased soon by the installation of 210-foot antennas at the DSN stations. As indicated in Figure 8a, the useful communication range of Pioneer is thereby extended to about 2 AU from the Earth so that missions of about two years duration can be flown, staying within 0.8 and 1.2 AU from the Sun. Missions such as these eventually occult the Sun and allow observations of the electron densities in the solar corona to be made.

By adding a despun reflector to focus the antenna pattern into a 5 by 20 degree beam aimed at the Earth, the region of space that can be explored with the 210-foot ground antenna is shown in Figure 8b. In order to extend Pioneer flights further out from the Sun, to about 1.4 AU, without using batteries, the transmitter power would be decreased to 4 watts and the resulting ranges are as shown in Figure 8b. With these two modifications, substantial data rates would be achievable anywhere between Venus and Mars orbits.

As an example of a mission utilizing this extended communication capability, Figure 9 shows two typical solar monitor missions, plotted in Earth-fixed coordinates. The spacecraft spends approximately one year in each loop, during which time observations can be made of solar phenomena before rotation of the Sun has brought these disturbances into sight of Earth observers. By choosing higher-velocity trajectories, these loops can be moved to a position 90 degrees trailing the Earth, still near Earth's orbit. The spacecraft would spend approximately 18 months in nearly the same relative position with respect to the Earth. This might prove useful for a solar flare early-warning system.

By adding appropriate filters over the solar array, solar array temperatures could be kept low enough for the spacecraft to reach about 0.4 AU from the Sun. Figure 10 shows two typical missions, plotted in Earth-fixed coordinates, utilizing a deeper penetration of the solar atmosphere corresponding to different launch velocities. Solar occultations to determine electron density measurements in the solar corona could be achieved in less than one year on trajectory B. The limits indicated in Figure 10 are those of the present Pioneer with the 210-foot ground station antenna without the transmitter or reflector modifications mentioned above.

Another improvement that might be made to extend Pioneer's range of applications would be to add a midcourse velocity correction engine. Studies of interplanetary trajectories have shown that by making the appropriate velocity correction along the two directions of spin axis normally available during flight (namely, along the Step I orientation direction perpendicular to the Sun line, and along the Step II orientation direction, normal to the ecliptic), any guidance error made by the launch vehicle at injection can be corrected. The magnitude of the error that can be removed is controlled primarily by the fuel carried on the spacecraft, the time of correction, and the sensitivity of the particular trajectory flown. The final accuracy attained is controlled by the ground tracking accuracy, the number of correction maneuvers performed, and the sophistication of control of the engine itself. The modification could consist of adding a monopropellant hydrazine propulsion system which could fire in either of two directions, parallel to the spin axis, with a low thrust. This approach is quite similar to that already used or being developed for positioning of Earth-orbiting satellites. Having such a velocity correction system and the other modifications mentioned above would then allow Pioneer to fly within several thousand miles of any of the inner planets, asteroids, and certain comets, and at the same time monitor the properties of interplanetary space during the transit phase.

References

1. Frank A. Brown, Jr., "Biological Clocks and the Fiddler Crab", Scientific American, April 1954, Vol. 190, No. 4.

2. Jürgen Aschoff, "Circadian Rhythms in Man", Science, 11 June 1965, Vol. 148.

ILLUSTRATIONS

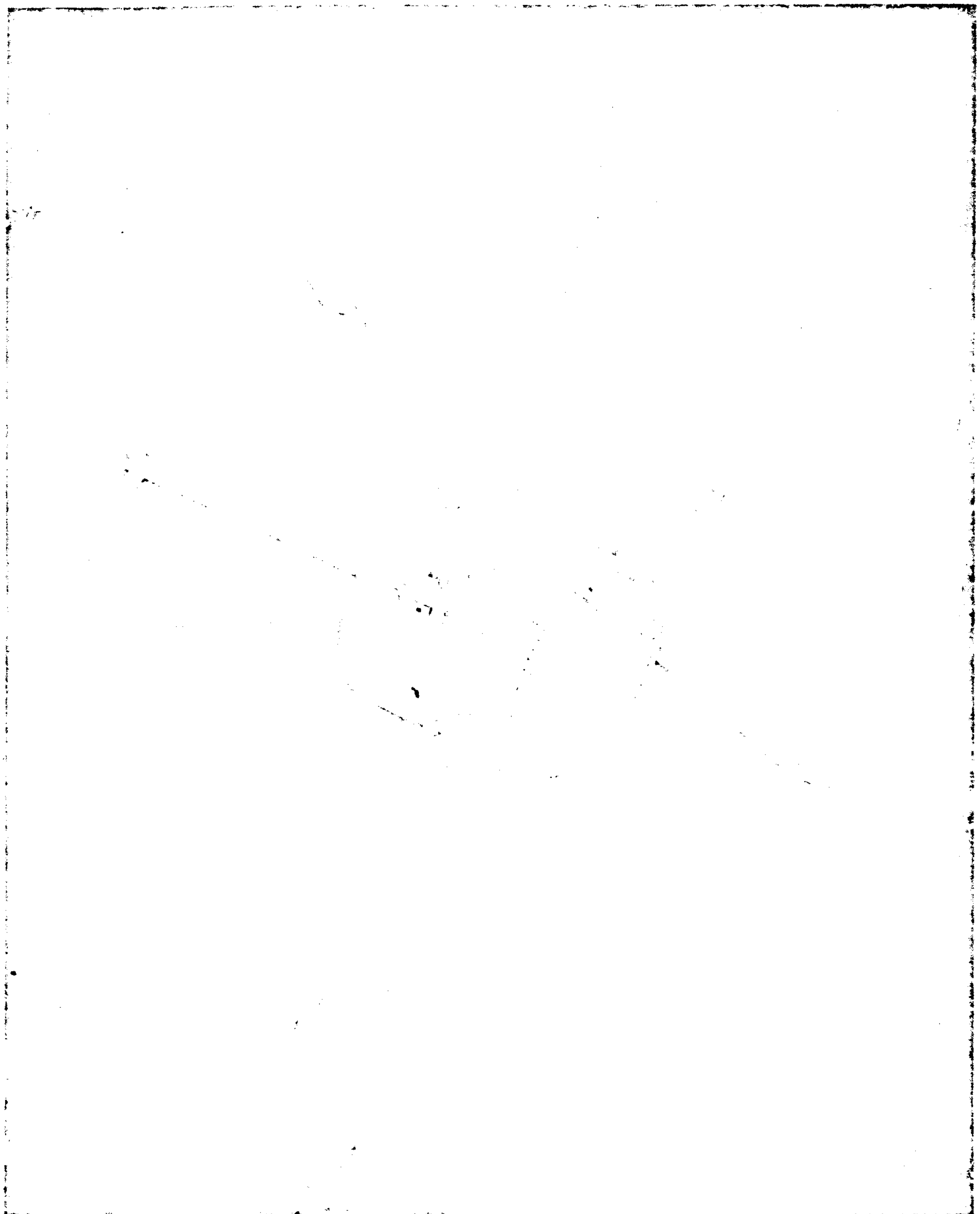


Figure 1. Pioneer 6 Spacecraft

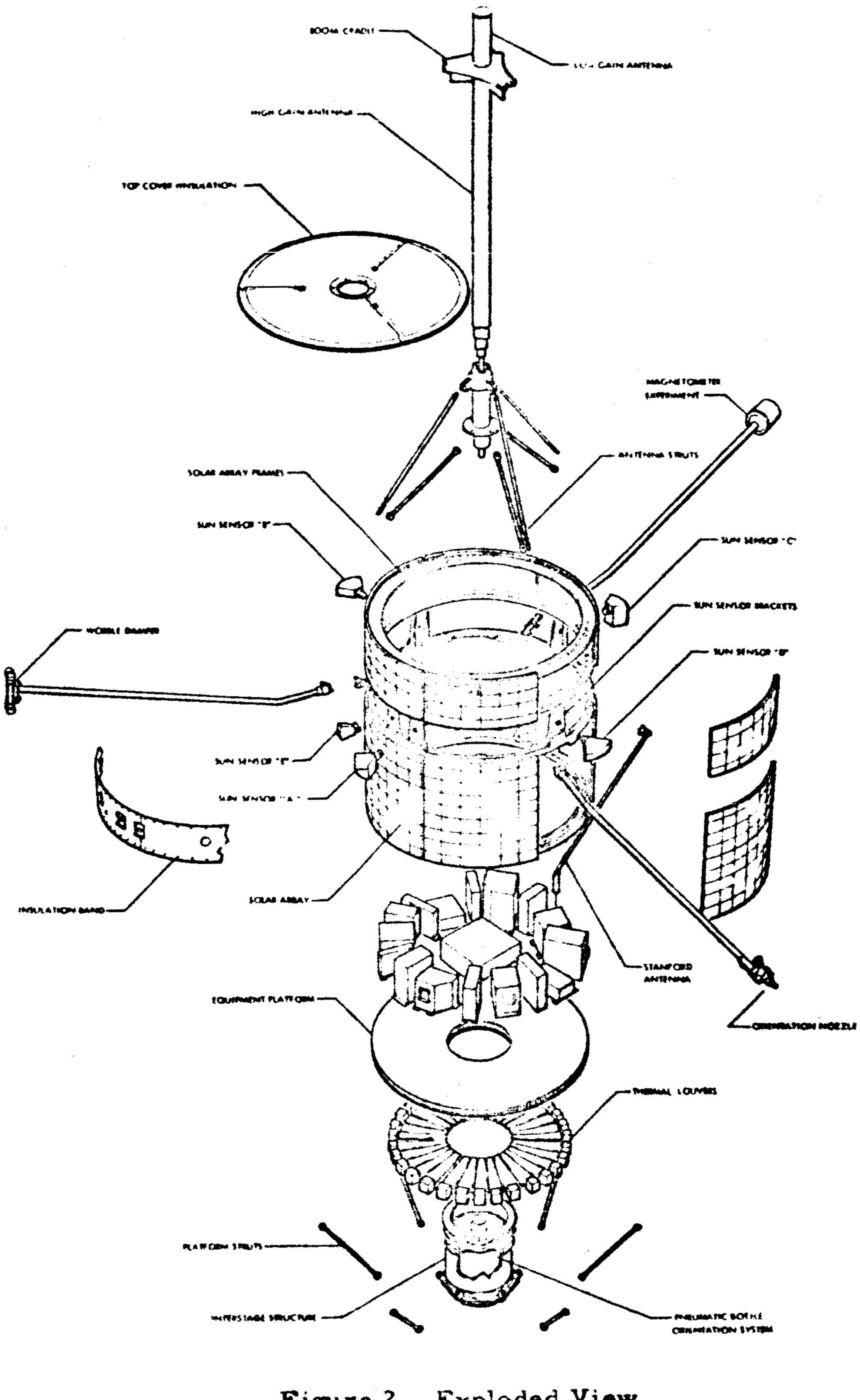


Figure 2. Exploded View

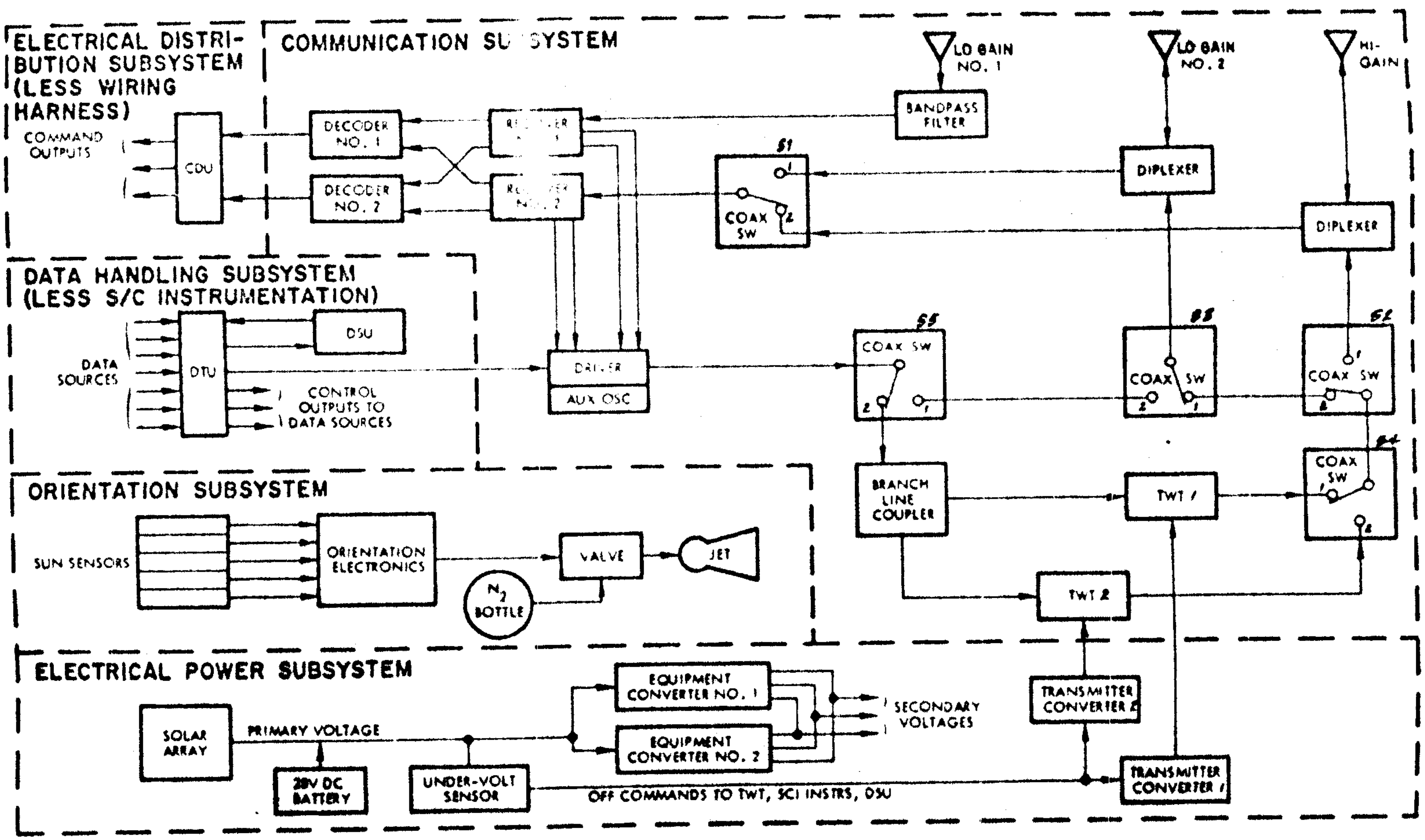


Figure 3. Simplified Functional Block Diagram

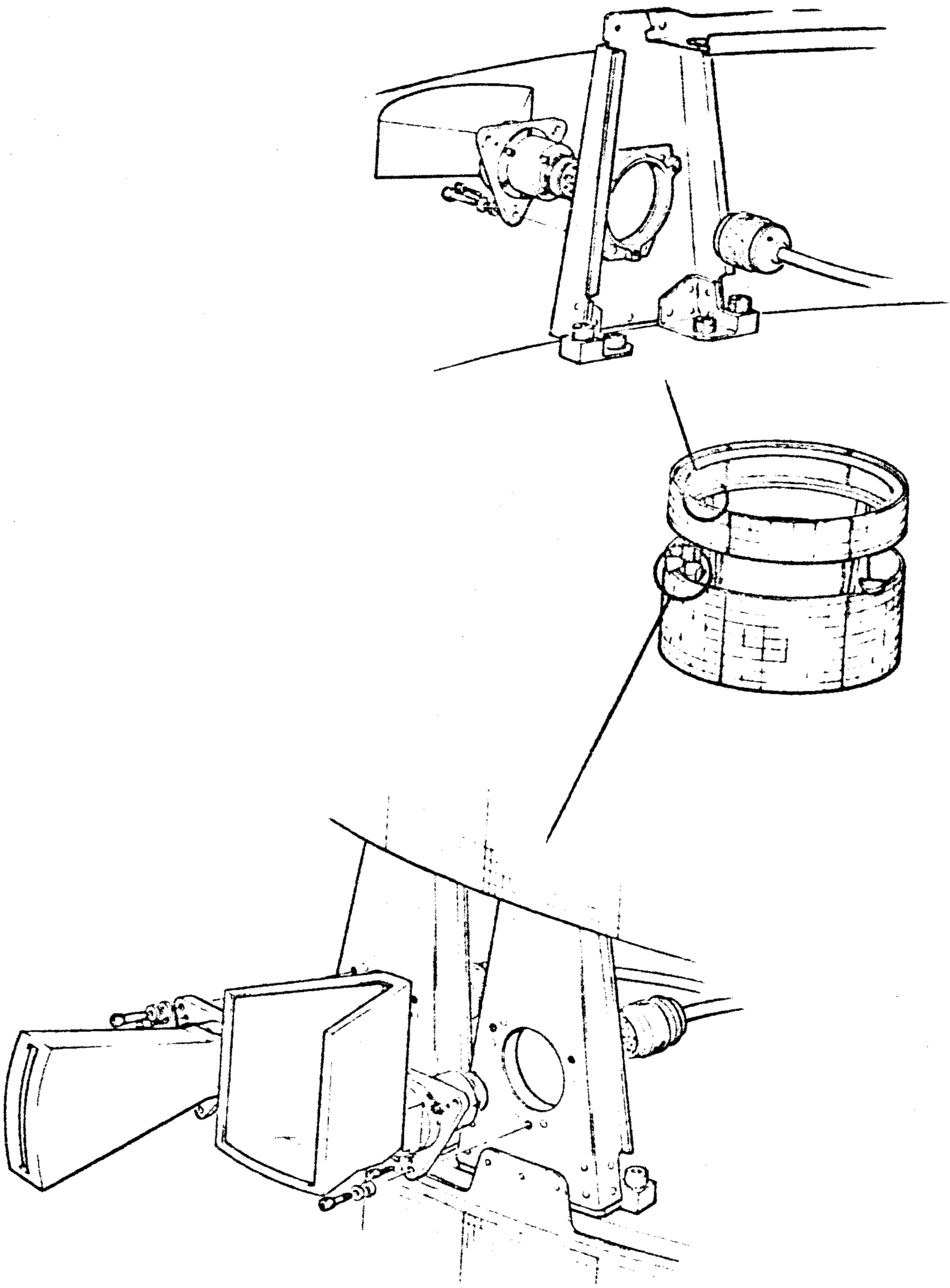


Figure 4. Orientation Sun Sensors

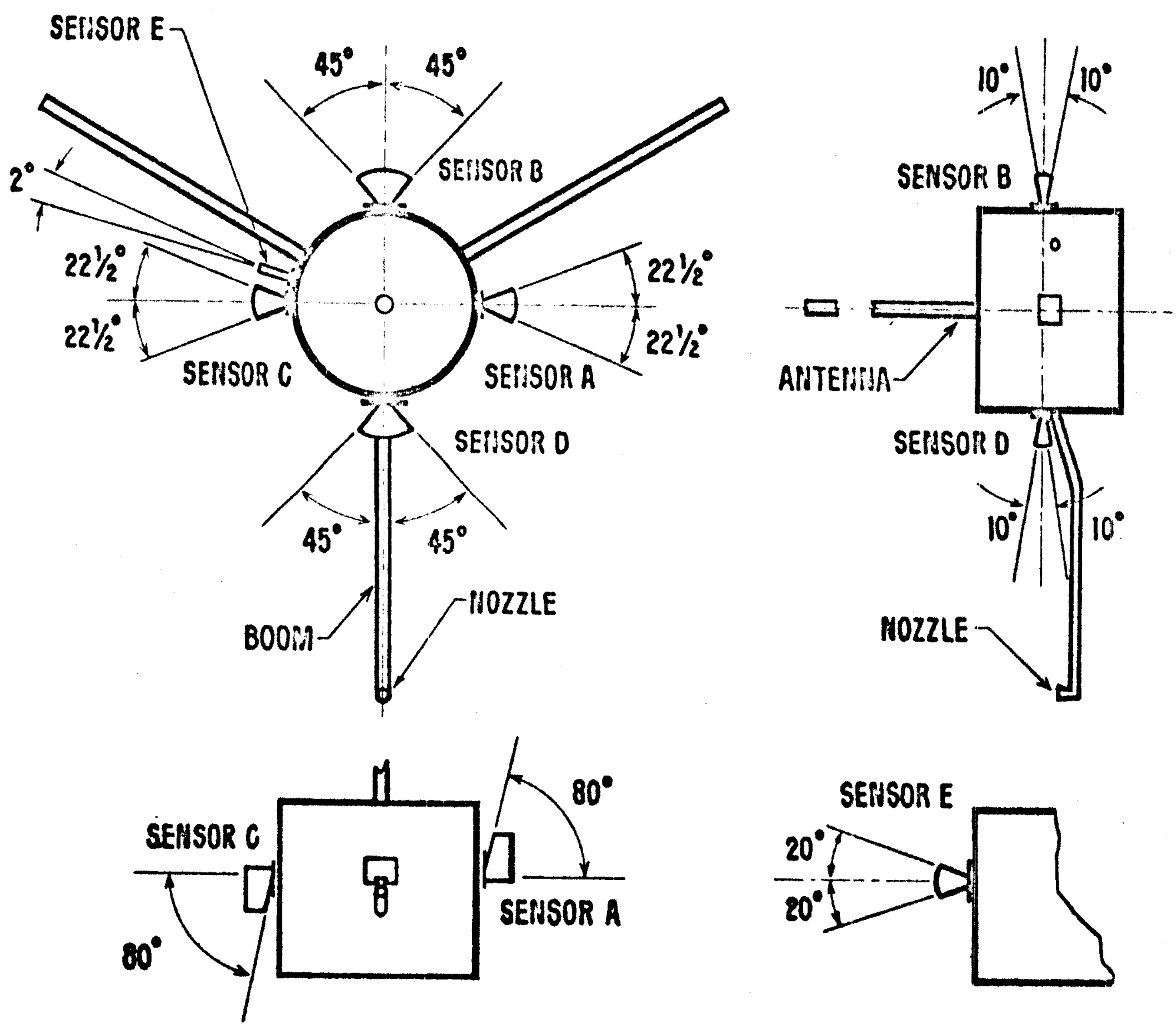


Figure 5. Sun Sensor Locations and Fields of View

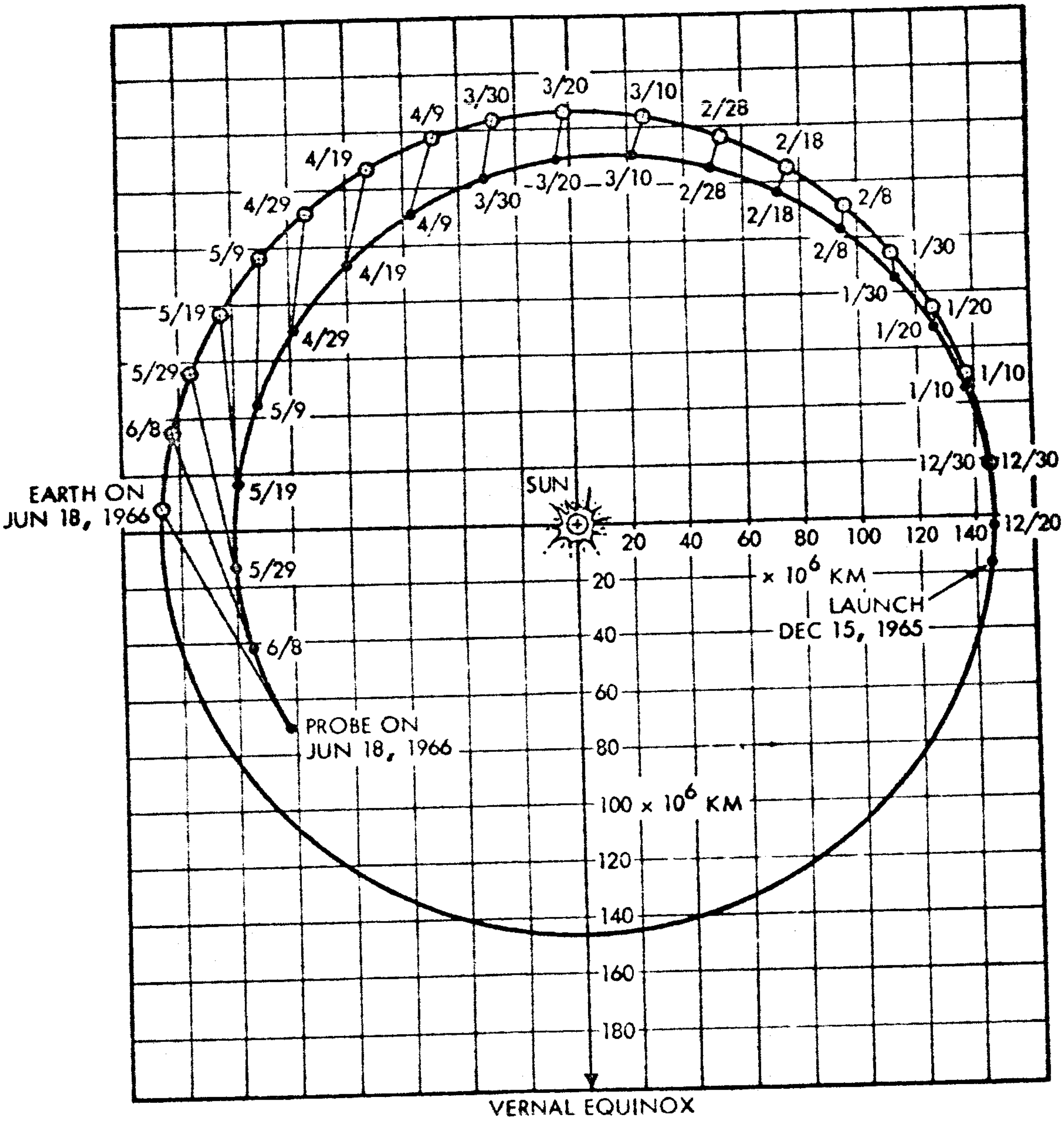


Figure 6. Pioneer 6 Trajectory

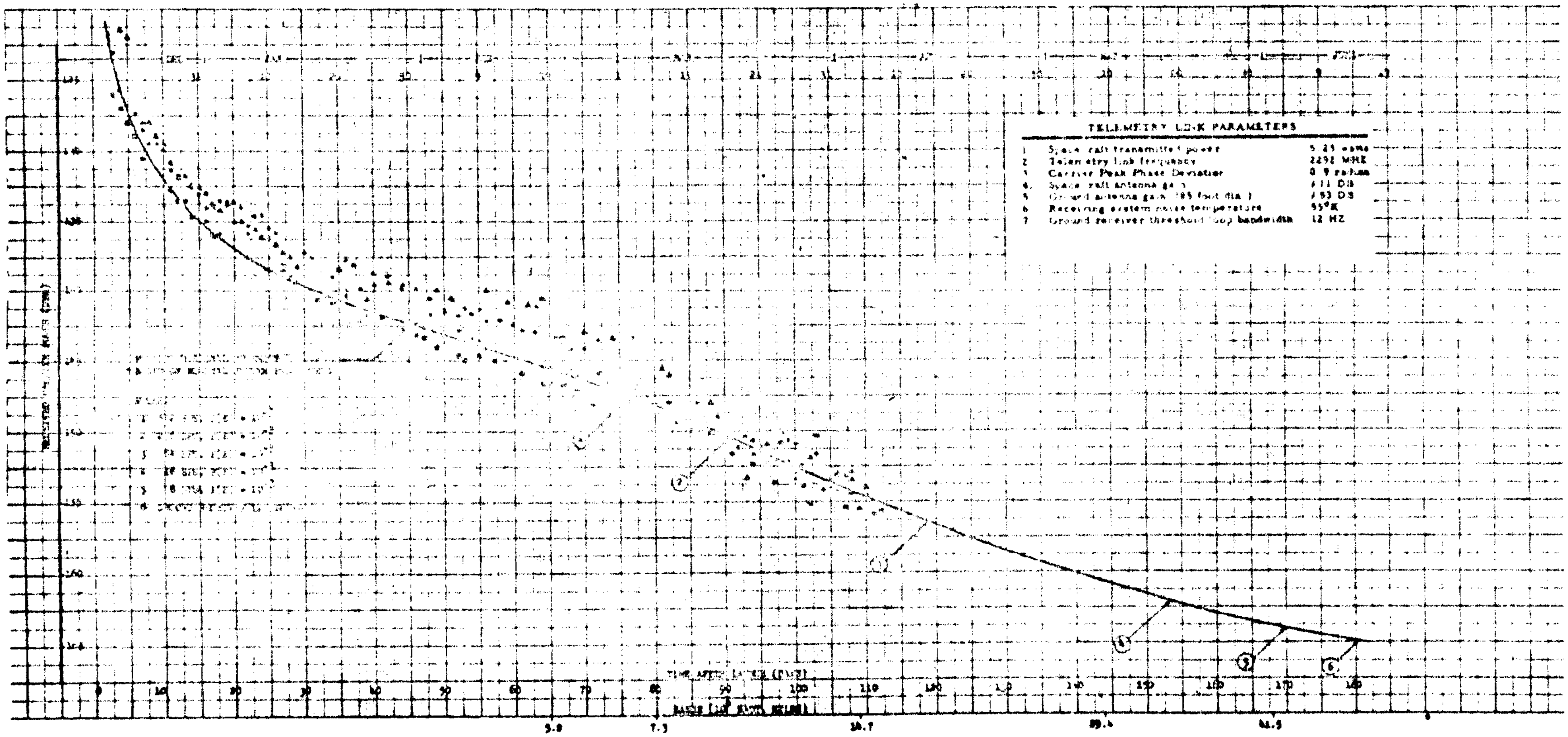
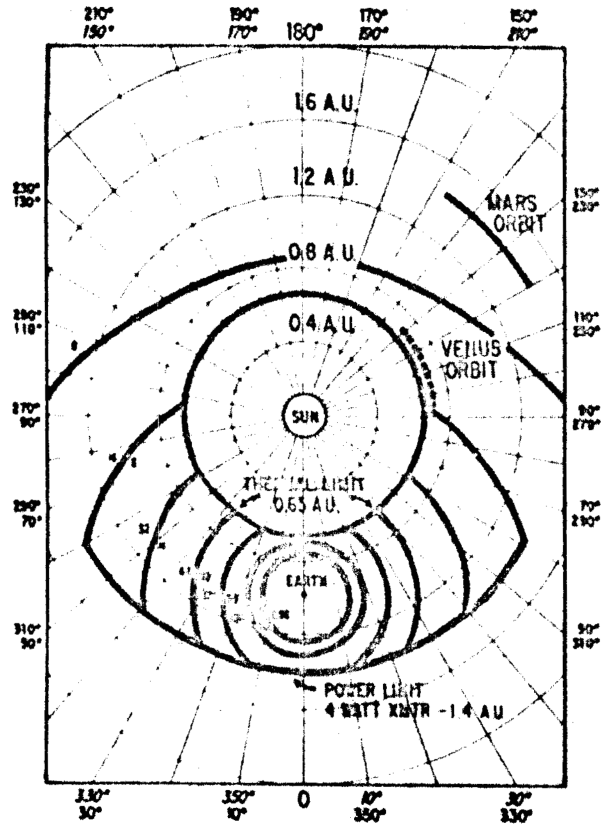
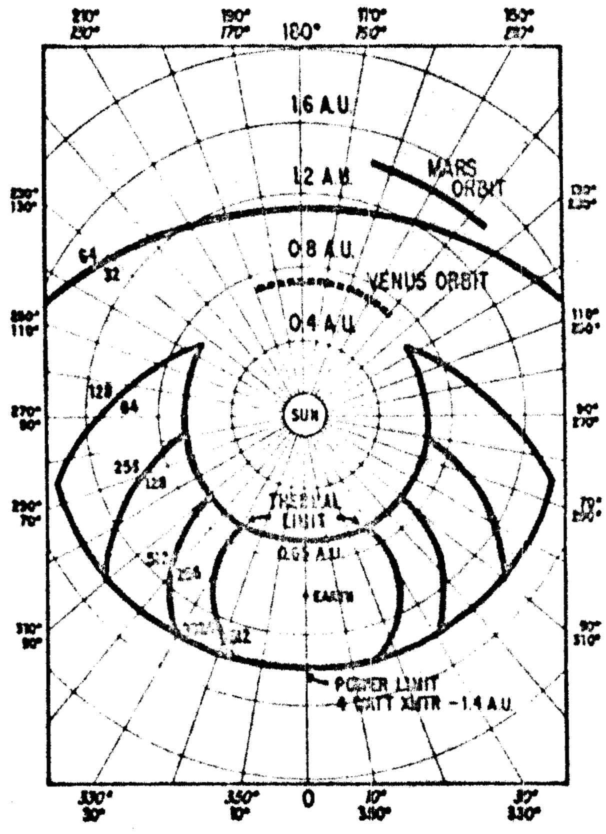


Figure 7. Pioneer 6 Received Carrier Signal Strength at Ground Receiver versus Time



a

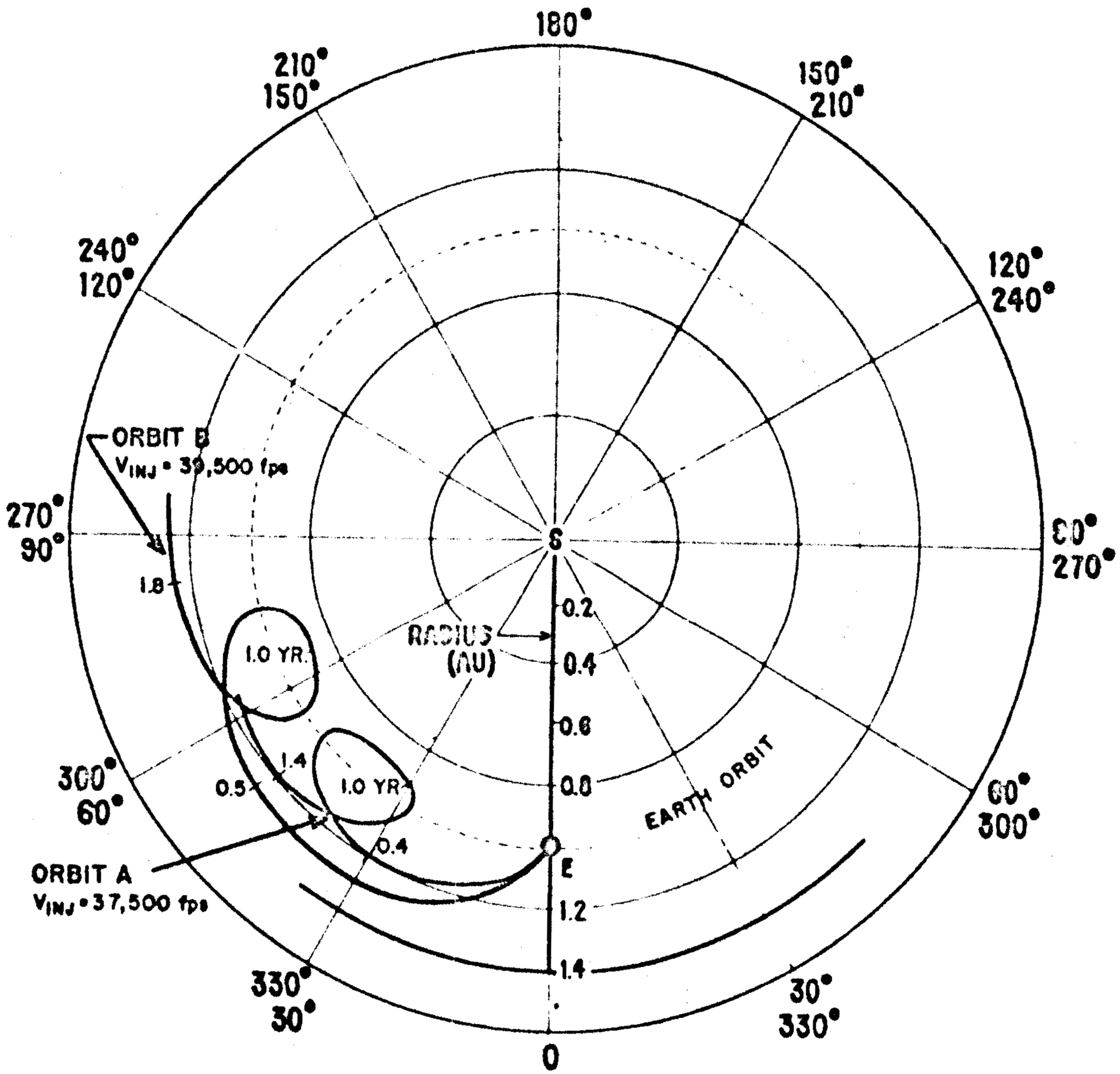


b

1. 22db SPACECRAFT ANT. + 85' GRD ANT. OR 11db SPACECRAFT ANT. + 210' GRD ANT.
2. LARGER NUMBER ON RANGE CURVE IS TRANSMISSION RATE WITH 8 WATT TRANSMITTER

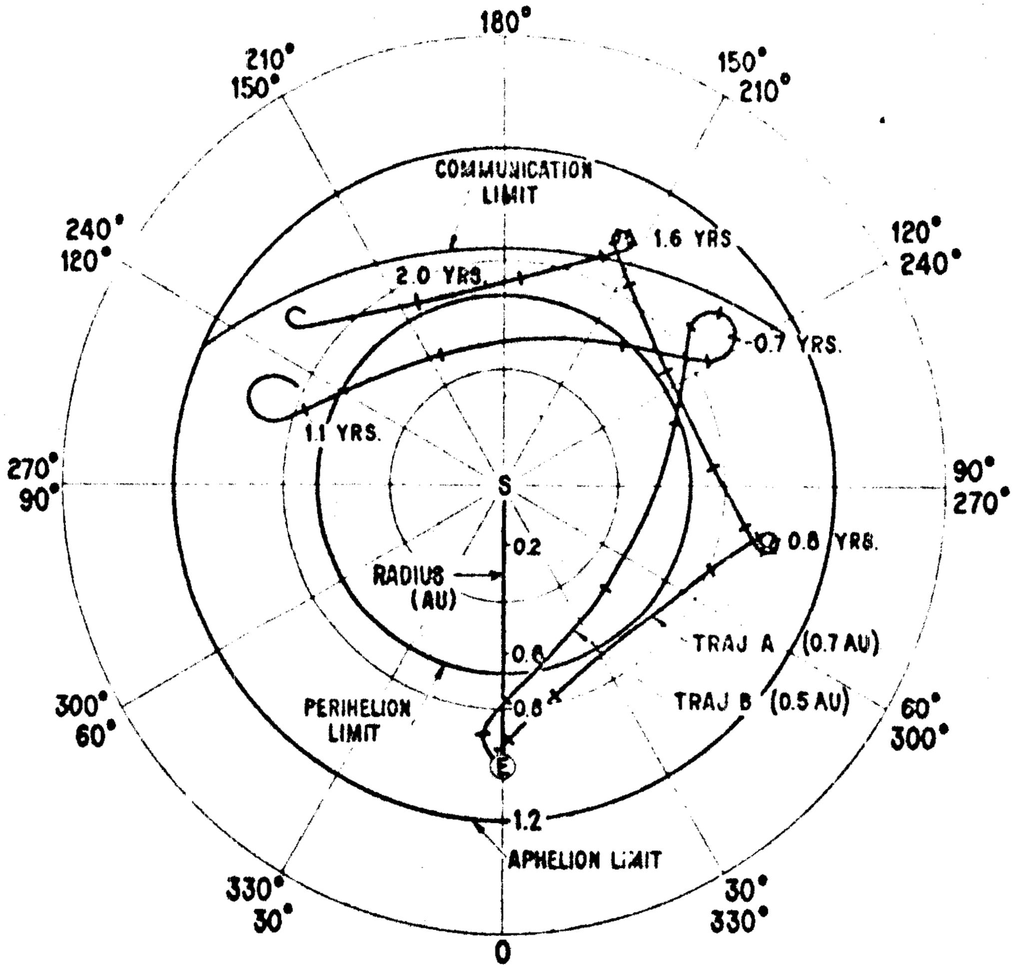
1. 22db SPACECRAFT ANTENNA
2. 210 FOOT GROUND ANTENNA
3. LARGER NUMBER ON RANGE CURVE IS TRANSMISSION RATE WITH 8 WATT TRANSMITTER

Figure 8a and 8b. Communication Performance



PRESENT BOOSTER: ORBIT A
ATLAS & TE 364 : ORBIT B

Figure 9. Typical Solar Monitor Trajectories



	TRAJ A	TRAJ B
$V_{INJ.}$	37,500 FPS	41,000 FPS
TIME OF OPPOS.	1.91 YR.	0.963 YR.
BOOSTER	TAD	THOR/DELTA/TE 364

Figure 10. Typical Solar Probe Trajectories (0.7 to 0.5 AU)